c6

NACA TN 3997

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3997

A NOTE ON THE EFFECT OF HEAT TRANSFER ON PEAK PRESSURE RISE ASSOCIATED WITH SEPARATION OF TURBULENT BOUNDARY LAYER ON A BODY OF REVOLUTION (NACA RM-10) AT A MACH NUMBER OF 1.61

By K. R. Czarnecki and Archibald R. Sinclair

Langley Aeronautical Laboratory
Langley Field, Va.

Washington
April 1957

LIBRARY COPY
APR 29 1957
LANGLEY AERONAUTICAL LABORATORY
LIBRARY, NACA
LANGLEY FIELD, VIRGINIA

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3997

A NOTE ON THE EFFECT OF HEAT TRANSFER ON PEAK PRESSURE RISE ASSOCIATED WITH SEPARATION OF TURBULENT BOUNDARY LAYER ON A BODY OF REVOLUTION (NACA RM-10) AT A MACH NUMBER OF 1.61

By K. R. Czarnecki and Archibald R. Sinclair

SUMMARY

An investigation has been made to determine the effect of heat transfer on the peak pressure rise associated with the separation of a turbulent boundary layer on a body of revolution (NACA RM-10) at a Mach number of 1.61. Tests were made over a Reynolds number range from 11.6×10^6 to 34.8×10^6 and with 0° to 120° F of cooling, which corresponds to a ratio of model-wall temperature to stagnation temperature of 0.96 (zero heat transfer) to 0.75. The stagnation temperature was approximately 570° F absolute. Boundary-layer separation was induced by means of forward-facing steps or collars at the base of the model and changes in heat transfer were obtained by cooling the model. The peak pressure rise was determined from shock angles measured from schlieren photographs.

The results indicated little or no effect of heat transfer on the shock angles associated with separation and, hence, on the peak pressure rise required to separate a turbulent boundary layer. The technique of using shock angles to determine the peak pressure rise for separation gave average results that were in good agreement with those of previous investigations in which measured pressure distributions were employed.

INTRODUCTION

In recent years a number of investigations have been made of the interactions of the shock and boundary layer at supersonic speeds. None of these investigations, however, have considered the effect of heat transfer. Inasmuch as a large proportion of the flights made at supersonic speeds are made without achieving thermal equilibrium, a knowledge of the

effects of heat transfer on the interaction of the shock and boundary layer would be of interest. Such knowledge would be of value, for example, in the estimation of spoiler effectiveness, hinge moments of flap-type controls when separation is present, and inlet and diffuser performance.

As a consequence of this need, a preliminary investigation has been made of the effect of boundary-layer cooling on the peak pressure rise required to separate the turbulent boundary layer near the base of a parabolic body of revolution (NACA RM-10). The tests were made at a Mach number of 1.61 over a Reynolds number range from 11.6×10^6 to 34.8×10^6, based on the distance from the model nose to the flow-separation point and free-stream conditions, and with 0° to 120° F of cooling, which corresponds to a ratio of model-wall temperature to stagnation temperature of 0.96 (zero heat transfer) to 0.75. The stagnation temperature was approximately 570° F absolute. Boundary-layer separation was obtained by means of forward-facing steps or collars at the rear of the model and the peak pressure rise for separation was determined from shock angles measured from schlieren photographs.

SYMBOLS

Δp	static-pressure rise across separation shock
q_1	dynamic pressure based on local conditions just ahead of flow separation
$\frac{\Delta p}{q_1}$	peak-pressure-rise coefficient across separation shock for forward-facing step
ΔT	average amount of model cooling, $^\circ$F
T_w	average model-wall temperature, $^\circ$F absolute
T_o	stagnation temperature, $^\circ$F absolute
$\frac{T_w}{T_o}$	ratio of model-wall temperature to stagnation temperature
R	Reynolds number based on distance from model nose to separation point and free-stream conditions
M	Mach number

APPARATUS AND TESTS

Wind Tunnel

The tests were made in the Langley 4- by 4-foot supersonic pressure tunnel which is described in reference 1. Calibration of the test-section flow at $M = 1.61$ indicates a Mach number variation of about ± 0.01 and no significant flow irregularities in the stream flow direction.

Model and Techniques

Pertinent dimensions, construction details, and thermocouple locations are given in figure 1 for the NACA RM-10 model without fins. The body has a parabolic-arc profile with a basic fineness ratio of 15. The pointed stern has been cut off at 81.25 percent of the original length, however, so that the actual body has a length of 50 inches and a maximum diameter of 4.096 inches. A detailed description of the model and the cooling technique is given in reference 1.

Details of the forward-facing steps or collars used to force boundary-layer separation and of their location with reference to the basic model are also given in figure 1. A photograph of the model with the large collar installed is presented as figure 2. The ratios of the heights of the steps to the boundary-layer thickness that existed at the base of the body without the steps were roughly 1:1 and 2:1 for the small and large steps, respectively, for the range of Reynolds numbers investigated. Boundary-layer separation occurred at approximately the 87- and 92-percent body stations for the two collars, where the pressure gradient on the body was nearly zero experimentally (ref. 2).

The test procedure consisted of adjusting the tunnel pressure to provide the desired Reynolds number and then cooling the model by injection of liquid carbon dioxide. When the model had cooled to a reasonably low temperature (that is, $\Delta T = -120^{\circ}$ F), the carbon dioxide was turned off and schlieren photographs and temperature distributions were taken at intervals as the model warmed to an equilibrium condition. When this cooling-heating cycle was completed, the tunnel pressure was adjusted to a new value and the procedure was repeated.

Range of Tests

Tests were made at $M = 1.61$ over a Reynolds number range from 11.6×10^6 to 34.8×10^6, based on the distance from the model nose to the point of flow separation and free-stream flow conditions. The cooling

range varied from $\Delta T = 0^\circ$ F to $\Delta T = -120^\circ$ F, which corresponds to a range of T_w/T_o from 0.96 (zero heat transfer) to 0.75. The stagnation temperature was approximately 570° F absolute.

Actually, tests were made to Reynolds numbers as low as 4×10^6, based on model length. With a laminar boundary layer at separation, however, the separation shock was so weak and the separation phenomena so diffused that it was impossible to identify the separation shock and, hence, to obtain any quantitative data. (See fig. 3 for typical schlieren photographs.) Consequently, the data in this report have been limited to conditions for which the boundary layer was known to be fully turbulent at or ahead of the separation point. In almost all cases the type of boundary layer at the separation point could be established from the schlieren negatives. The laminar boundary layer appears thinner and has a sharper boundary than the turbulent boundary layer. (Compare figs. 3, 4, and 5 ahead of boundary-layer separation point.)

Data Reduction

The peak-pressure-rise coefficients associated with boundary-layer separation were obtained by measuring the inclination of the separation shocks relative to the model surface and by computing the equivalent pressure rise through the shock for the existing local Mach number by use of three-dimensional-flow equations. (For details of peak-pressure-rise coefficients see ref. 3.) The pressure data of reference 2 were used to determine the local Mach numbers and the corresponding local dynamic pressures. Only the straight-line part of the separation shock outside the boundary layer was used to determine the shock angles. There is a question about the absolute accuracy of the peak-pressure-rise coefficients determined by this technique. The accuracy of the incremental changes due to heat transfer, however, is believed to be sufficient for this purpose.

RESULTS AND DISCUSSION

Some typical schlieren photographs of the boundary-layer separation phenomena with a turbulent boundary layer are presented as figures 4 and 5. The effects of changes in Reynolds number at zero heat transfer are shown in figure 4, and the effects of heat transfer (boundary-layer cooling) at constant Reynolds number are shown in figure 5. General inspection of these and other schlieren photographs taken during these tests reveals that the effects of Reynolds number and heat transfer, if they are existent, are small.

The effect of Reynolds number on the peak-pressure-rise coefficient associated with the separation of the turbulent boundary layer is indicated

in figure 6. The data show considerable scatter; the variation in the peak-pressure-rise coefficient with R cannot be determined exactly but it is indicated to be small. No effect due to collar size is apparent. The average value of the coefficient of about 0.44 is equivalent to a peak-pressure-rise ratio (ratio of static pressure behind the shock to that ahead of the shock) of about 1.85 (local Mach number = 1.66). This value is in fairly good agreement with some results obtained by means of pressure distributions on spoilers in unpublished two-dimensional tests made in the Langley 4- by 4-foot supersonic pressure tunnel and with the experimental results compiled in reference 4. Hence, the present technique of determining the peak-pressure-rise coefficient for separation of the turbulent boundary layer appears to give results of the right order of magnitude, since the separation phenomena may be expected to be basically similar in either two- or three-dimensional flow.

The effect of heat transfer (boundary-layer cooling) on the peak-pressure-rise coefficient associated with the separation of the turbulent boundary layer is presented in figure 7. Data for both collars and all test Reynolds numbers are included. There is little or no effect on $\Delta p/q_1$ due to boundary-layer cooling for either the large or small collar and at any Reynolds number for as much as 120° F of cooling (T_w/T_o range from 0.96 (zero heat transfer) to 0.75).

SUMMARY OF RESULTS

An investigation has been made to determine the effect of heat transfer (boundary-layer cooling) on the peak pressure rise associated with the separation of a turbulent boundary layer on a body of revolution (NACA RM-10). The tests were made at a Mach number of 1.61 over a Reynolds number range, based on the distance from the model nose to the flow separation point, from 11.6×10^6 to 34.8×10^6 for a ratio of model-wall temperature to stagnation temperature of 0.96 (zero heat transfer) to 0.75. The following results were indicated:

1. Heat transfer had little or no effect on the shock angles associated with separation and, hence, on the peak pressure rise required to separate a turbulent boundary layer.

2. The technique of using shock angles to determine the peak pressure rise for separation gave average results that were in fairly good agreement with those of previous investigations in which pressure distributions were employed.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., February 6, 1957.

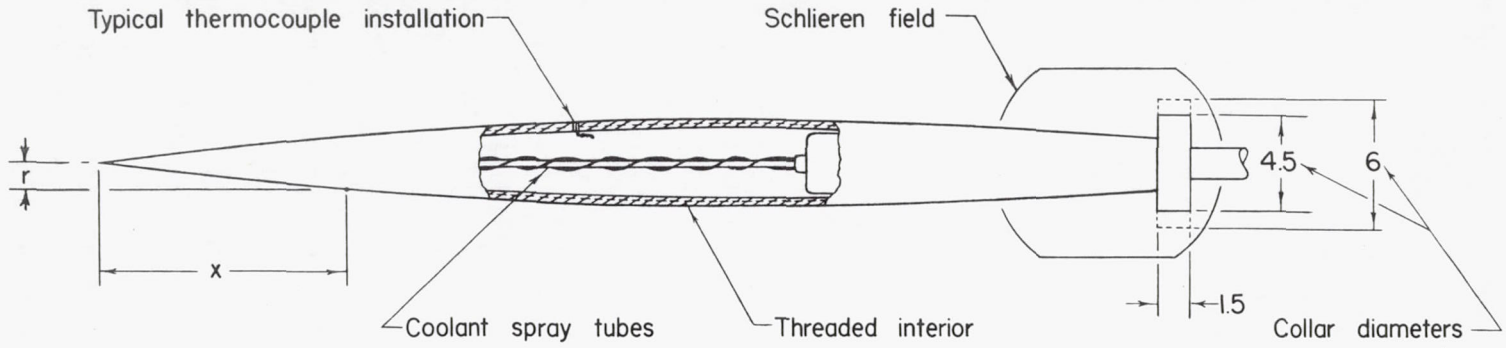
REFERENCES

1. Czarnecki, K. R., and Sinclair, Archibald R.: An Investigation of the Effects of Heat Transfer on Boundary-Layer Transition on a Parabolic Body of Revolution (NACA RM-10) at a Mach Number of 1.61. NACA Rep. 1240, 1955. (Supersedes NACA TN's 3165 and 3166.)

2. Czarnecki, K. R., and Marte, Jack E.: Skin-Friction Drag and Boundary-Layer Transition on a Parabolic Body of Revolution (NACA RM-10) at a Mach Number of 1.6 in the Langley 4- by 4-Foot Supersonic Pressure Tunnel. NACA RM L52C24, 1952.

3. Bogdonoff, S. M., and Kepler, C. E.: Separation of a Supersonic Turbulent Boundary Layer. Jour. Aero. Sci., vol. 22, no. 6, June 1955, pp. 414-424, 430.

. Reshotko, Eli, and Tucker, Maurice: Effect of a Discontinuity on Turbulent Boundary-Layer-Thickness Parameters With Application to Shock-Induced Separation. NACA TN 3454, 1955.



Body profile equation: $r = 0.1333x - 0.00217x^2$

Model length: 50.0

Maximum diameter: 4.096

Thermocouple locations		
Station	Number	Spacing
3.0	2	180°
12.6	2	180°
22.4	4	90°
32.0	2	180°
37.1	2	180°
46.0	2	180°

Figure 1.- NACA RM-10 model and apparatus for cooling. All dimensions are in inches unless otherwise indicated.

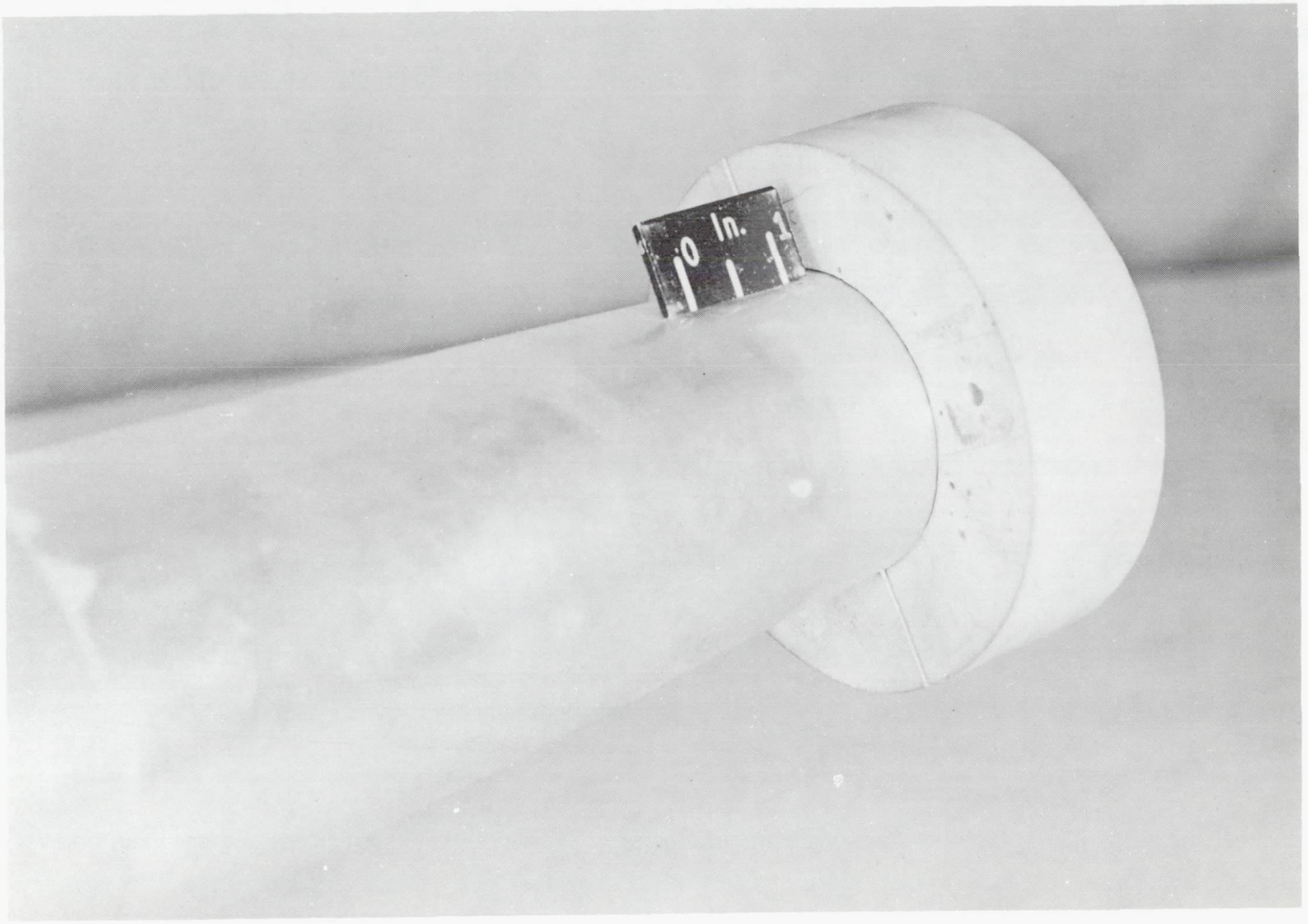
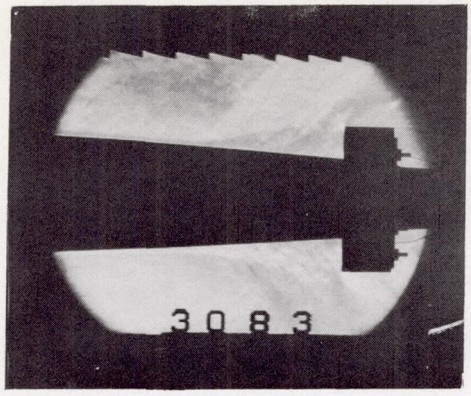
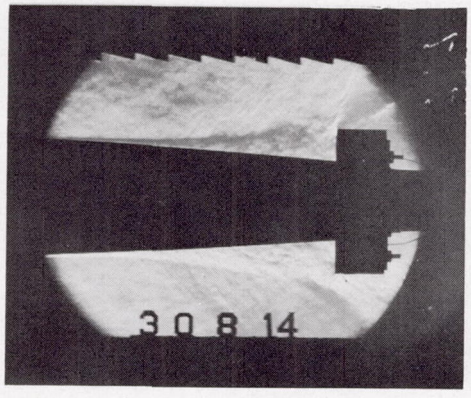


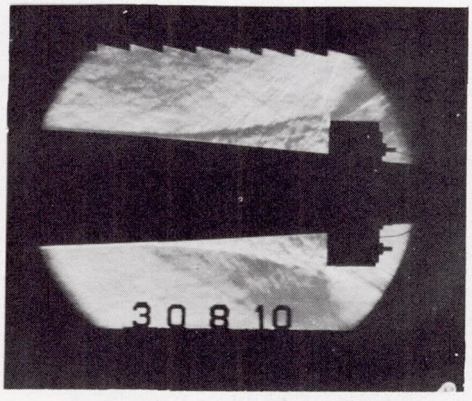
Figure 2.- Large collar mounted at base of NACA RM-10 missile. L-94517



$R=5.8 \times 10^6, \Delta T=-106°$



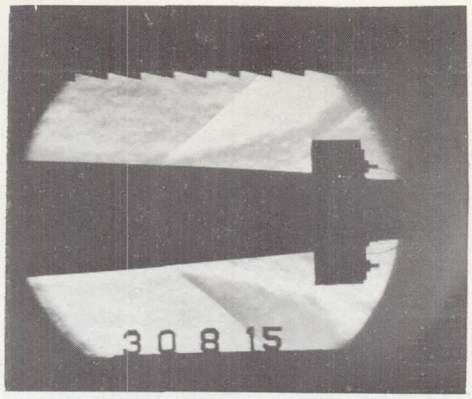
$R=11.6 \times 10^6, \Delta T=-12°$



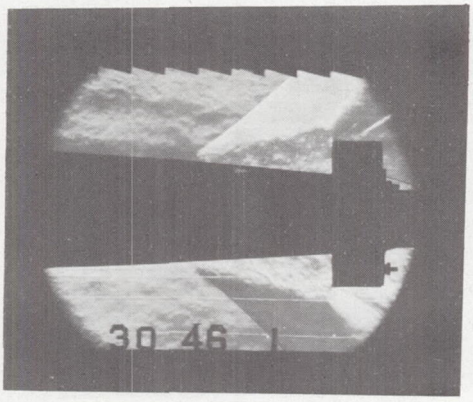
$R=11.6 \times 10^6, \Delta T=-81°$

L-57-164

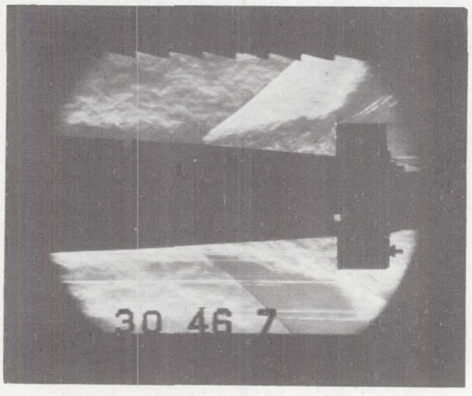
Figure 3.- Typical schlieren studies of boundary-layer separation with laminar boundary layer.



$R=17.4\times10^6$



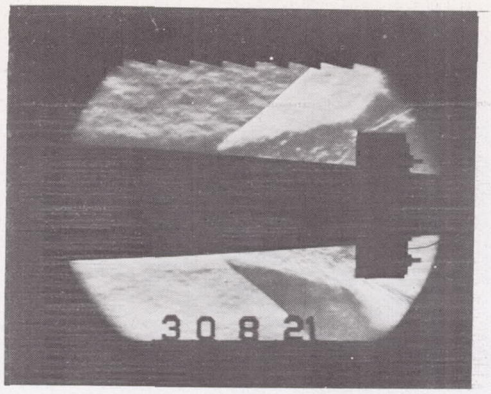
$R=27.8\times10^6$



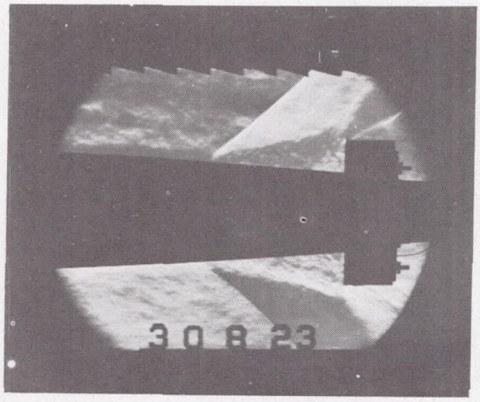
$R=34.8\times10^6$

L-57-165

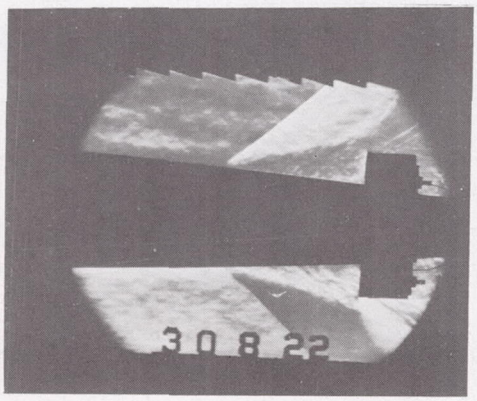
Figure 4.- Schlieren studies showing effect of Reynolds number on separation of the turbulent boundary layer. Small collar; zero heat transfer.



$\Delta T = 0°$



$\Delta T = -51°$



$\Delta T = -120°$

L-57-166

Figure 5.- Schlieren studies showing effect of cooling on separation of the turbulent boundary layer. Small collar; $R = 23.2 \times 10^6$.

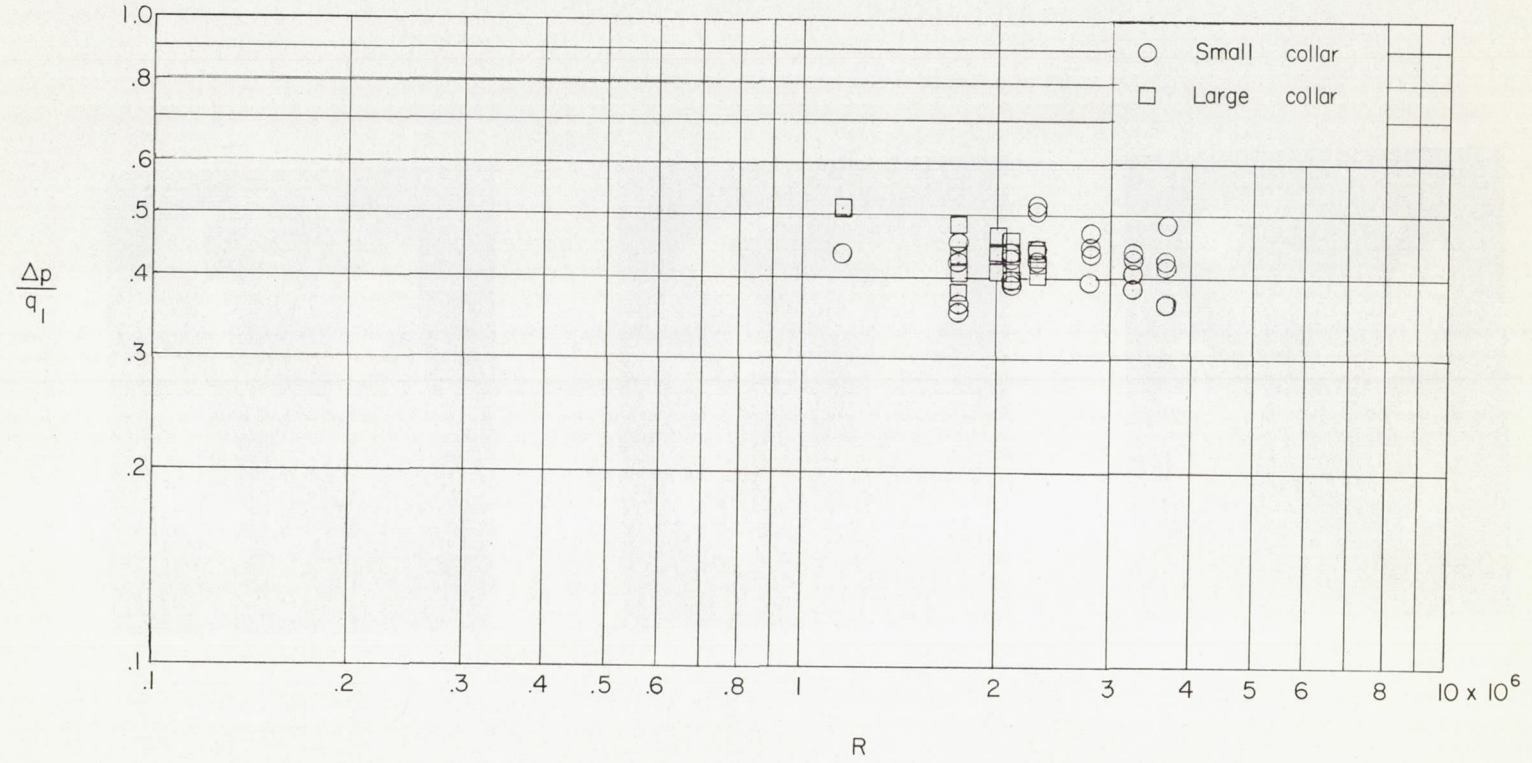


Figure 6.- Effect of Reynolds number on peak-pressure-rise coefficient associated with the separation of a turbulent boundary layer with zero heat transfer.

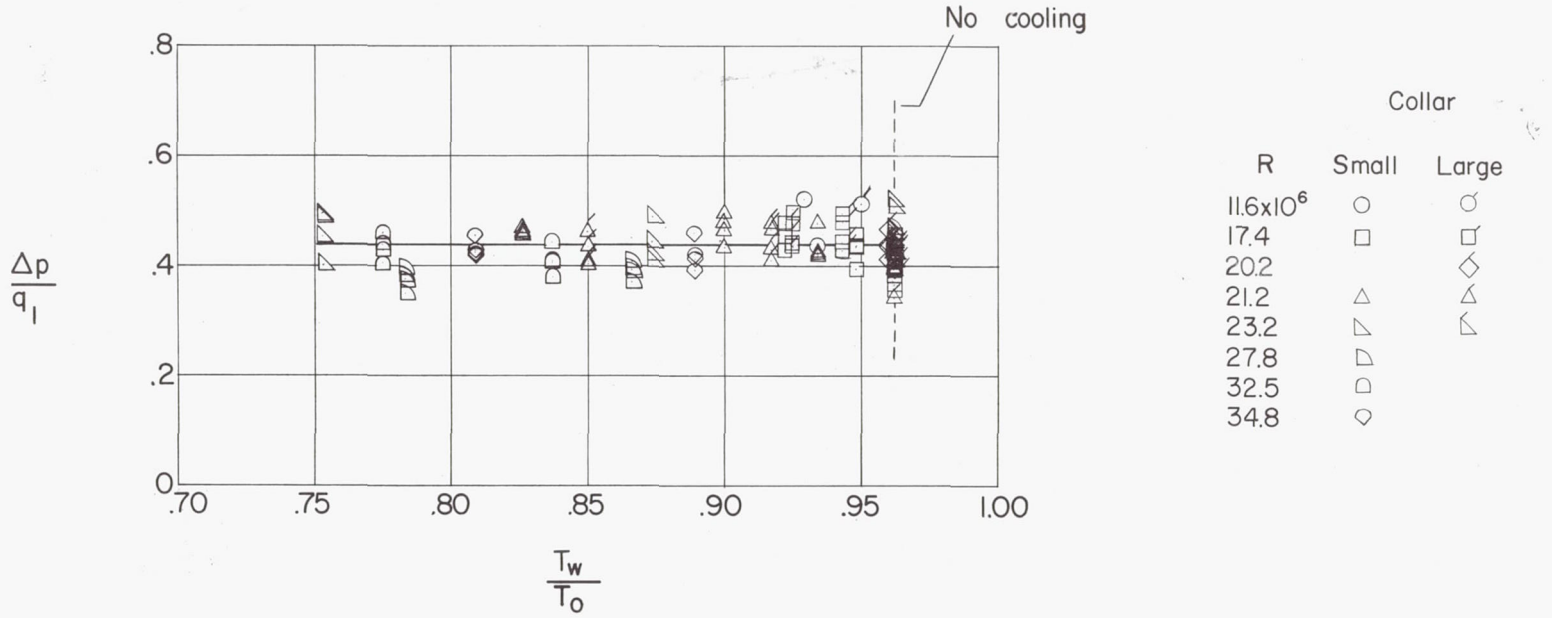


Figure 7.- Effect of cooling on peak-pressure-rise coefficient associated with the separation of a turbulent boundary layer.

NACA - Langley Field, Va.